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NLR-TP-2015-472 | November 2015

Innovative Small Launcher

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Innovative Small Launcher

Problem area

The market for small satellites is expected to increase substantially in the coming years, but there is little capacity to launch them affordably. No operational dedicated launcher for small satellites exists today. Small satellites, launched as secondary payloads, are entirely dependent on the constraints set by the primary payload, such as launch date and target orbit. Launch costs of less than €50,000 per kg of payload are required in order to directly compete with piggy-back ride shares. With a dedicated launcher a higher cost per kg can be accepted for payloads which need to be delivered timely and accurately to a desired orbit.

Description of work

The SMILE project aims at a combined research approach into a new innovative European launcher for an emerging market of small satellites up to 50 kg using a multidisciplinary design and optimisation approach strengthened by the demonstration of critical technologies for cost-effective solutions and complemented with the design of a European-based launch capability from Andøya (Norway).

Critical launcher technologies in various expertise areas will be developed in SMILE, but this paper focusses on the rocket engine developments and their impact on cost reduction and design since the engines are the most critical and expensive parts of a launcher. For the rocket engines, both hybrid engines and reusable liquid engines are assessed.

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Results and conclusions

For the propulsion systems there is a significant difference in terms of a cost effective design approach. Whereas the hybrid case focusses on volume production and inherent low development and operational cost, the liquid case focusses on increased lifetime to enable reusability.

Applicability

The combination of applied research on both the two propulsion technologies will allow the use of the right technology at the right place to offer the required performance at the lowest price possible. Ultimately, the choice of the propulsion system for all the stages of the rocket will be a trade-off between performance, launch objectives and cost.

GENERAL NOTE

This report is based on a presentation held at the 13th Reinventing Space Conference, Oxford, November 11th 2015.

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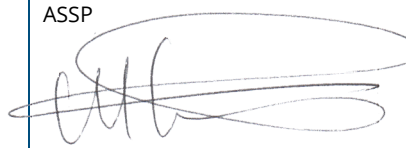
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Summary

The market for small satellites is expected to increase substantially in the coming years, but there is little capacity to launch them affordably. No operational dedicated launcher for small satellites exists today. Small satellites, launched as secondary payloads, are entirely dependent on the constraints set by the primary payload, such as launch date and target orbit. Launch costs of less than €50,000 per kg of payload are required in order to directly compete with piggy-back ride shares. With a dedicated launcher a higher cost per kg can be accepted for payloads which need to be delivered timely and accurately to a desired orbit.

A consortium of 13 companies and institutes are joining forces in a Horizon 2020 work programme to design a dedicated small launcher to be built in and launched from Europe. The project is called “SMall Innovative Launcher for Europe” (SMILE) and is currently in its preparation phase for the Grant Agreement with the European Commission. Kick-off is planned for 1/1/2016. The SMILE project aims at a combined research approach into a new innovative European launcher for an emerging market of small satellites up to 50 kg using a multidisciplinary design and optimisation approach strengthened by the demonstration of critical technologies for cost-effective solutions and complemented with the design of a European-based launch capability from Andøya (Norway). For the intended market, cost reduction is essential. One option to reduce cost is to apply reusability of one or more of the stages. Cost can also be reduced by applying commercial industry-grade components. Another means of cost reduction is through volume production. Finally, the production process can be optimized for cost, e.g. automated manufacturing for composite parts and 3D-printing for metallic parts. Critical launcher technologies in various expertise areas will be developed in SMILE, but this paper focusses on the rocket engine developments and their impact on cost reduction and design since the engines are the most critical and expensive parts of a launcher. For the rocket engines, both hybrid engines and reusable liquid engines are assessed.

Hybrid engines combine some of the advantages (simplicity, both in functioning and in hardware) of solid engines with those of liquid engines (inherent safety, throttling). The chosen combination of propellants (H₂O₂/HTPB) gives good performances on a wide range of mixture ratio, thus allowing a great versatility of the mission. Besides, it offers the advantage of being already available in industrial quantities, while being completely green (only CO₂ and H₂O produced). The engines and their propellants are also safe to handle (nontoxic constituents) and safe to operate (the two propellant ingredients stored separately). Those characteristics, coupled with a simple fluid system, will substantially reduce hybrid propulsion life cycle cost. In order to keep the price of the propulsion system as low as possible, reusability of components is a key feature leading to cost reductions through volume production and increased reliability through automated production. In that sense, a Unitary Motor is thought of as a building block that can be clustered to deliver the required thrust for a micro-launcher.

Liquid propulsion is a reliable technology which is favourable due to its flexibility as the engines can be throttled at a wide range and easily re-ignited. The combination of LOX/kerosene as green propellants seems to be very promising. Compared to other propellants, both oxidiser and fuel are low-cost, worldwide available and storable. The envisaged engine design is thereby based on ceramic materials. Due to their oxidation resistance, high specific strength and low thermal expansion behaviour at high

temperatures, such ceramic materials are specifically suited for liquid propulsion components. Applying fibre-reinforced ceramics, the material's characteristics can be further improved yielding into damage tolerant and reliable structures, being insensitive against thermo-shocks as well as thermal cycling. Compared to classical ITAR-controlled metal alloys (as the current main material for thrust chamber assemblies), the envisaged ceramic materials are lightweight and not subject to ITAR-controls. In combination with 3-D printed components and the potential use of CFRP (carbon-fibre reinforced plastics) housing structures, the engine's structural weight can thereby be significantly reduced. It is expected that a combination of LOX/kerosene operation in a clustered design with multiple sub-scaled engines based on ceramic materials and a transpiration cooling technique enables a considerably improved engine lifetime. This could indeed pave the way for prospective reusable liquid rocket propulsion.

The combination of applied research on both the two propulsion technologies will allow the use of the right technology at the right place to offer a launcher delivering the required performance at the lowest price possible. Ultimately, the choice of the propulsion system for all the stages of the rocket will be a trade-off between performance, launch objectives and cost.

Contents

1	Introduction	6
2	Smile Project	8
3	Hybrid Rocket Engine Technology	11
3.1	Current State of the Technology	11
3.2	The Unitary Motor	12
3.3	Hybrid Rocket Stage for a Micro-Launcher	15
4	Liquid Rocket Engine Technology	17
4.1	Ceramic Based Design	17
4.2	Engine Reusability	20
4.3	Envisaged SMILE Approach	21
5	Conclusions	22
6	References	23

1 Introduction

The new generation ARIANE 6 and VEGA C launchers will guarantee Europe's independent access to space for the high-end market of satellites in terms of mass and size with a competitive edge in the world market of launchers. These launchers, however, are significantly less attractive for smaller satellites. The initiative therefore addresses reliable, affordable, quick, and frequent access to space for the emerging market of small satellites up to 50 kg, fulfilling the needs from the European space Research and Technology Development (RTD) community as well as commercial initiatives to put satellites into specific LEO orbits within a preferred time window. Herewith a market niche is addressed, which is projected to grow significantly in the coming decades and presently lack the availability of a dedicated European launcher.

The market for small satellites is expected to increase substantially in the coming years, as shown in market analyses of among others SpaceWorks Enterprises Inc (SEI, Nano/ Microsatellite Market Assessment 2015, August 2014) and shown in Figure 1. The excellent prospects for the small satellite market are confirmed by EuroConsult (Prospects for the Small Satellite Market, Feb 2015) with an estimate of more than 500 small satellites (nanosats, microsats, and minisats) to be launched in the next five years. Currently, the U.S. is the most active country in small satellite deployment with almost half of the 620 satellites launched in the past 10 years with Europe as the second-largest region. Historical analysis suggests the current supply of launch vehicles will not sufficiently serve future nano/microsatellite market demand.

Nanosats and microsats nowadays have to share a ride on a large rocket for a primary customer, which often causes conflicts with respect to the timeline and the orbit properties. Now that smaller satellites become technologically more advanced and mature, a call for 'affordable' dedicated launches is expedient for small satellite operators.

This situation has led to several initiatives of small launchers for various payloads in the range of 1 to 150 kg: India (Reusable Launch Vehicle, ISRO), New Zealand (Electron, Rocket Lab Ltd.) and USA (SuperStrypi, Aerojet Rocketdyne; LauncherOne, Firefly, Virgin Galactic; Lynx, XCOR; ALASA, DARPA). But also within Europe, efforts are ongoing: France (Eole, CNES), Norway (North Star, Nammo/Andøya Space Centre), Spain (Arion, PLD Space), Switzerland (SOAR, S3) and UK (Skylon, Reaction Engines Ltd.).

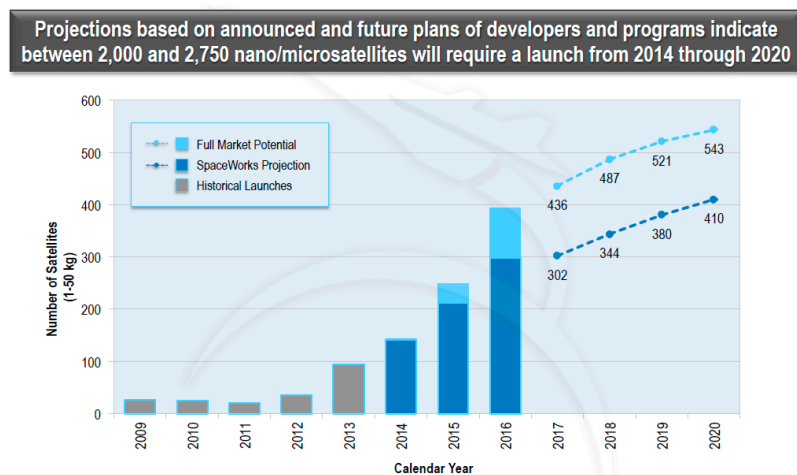


Figure 1 Future launch market for small satellites 1 - 50 kg (courtesy: SEI)

Although the above mentioned launch initiatives focus on the small satellite customer market, none of these focus on delivering the market's "sweet spot" to orbit and focus on specific payload launch ranges (e.g. 1-10 kg or 100kg+). Based on the market analyses the range up to 50 kg payload capacity can be considered the "sweet spot" for a small satellites launcher. Such a launcher will provide a proper launch capability for a single 50 kg satellite (i.e. commercial, scientific, and governmental) as well as for a flexible configuration of multiple smaller satellites (i.e. education, in-orbit demonstration) up to a total mass of about 50 kg. The above mentioned initiatives are in different states of development and are providing no launch services at this moment.

2 Smile Project

No operational dedicated launcher for small satellites exists today. Small satellites, launched as secondary payload, are entirely dependent of the constraints set by the primary payload, such as launch date and target orbit. Launch costs of less than €50,000 per kg of payload are required in order to compete directly with these piggy-back ride shares which are the current economically viable access to space for small satellites. With a dedicated launcher a higher cost per kg can be accepted for payloads which need to be delivered timely and accurately to a desired orbit. A shorter project schedule from concept to launch and better science are the arguments most commonly mentioned to support this. Hence, a consortium of 13 partners from 8 European countries are joining forces in a Horizon 2020 work programme to design a dedicated small launcher to be built in and launched from Europe. Together, the consortium coordinated by the Netherlands Aerospace Centre NLR covers all aspects of marketing, developing, and operating a cost-effective launcher with a well-balanced mix of companies, SMEs, and institutes.

The project is called “SMall Innovative Launcher for Europe”, SMILE, and is currently in its preparation phase for the Grant Agreement with the European Commission with a planned Kick-off date of 1/1/2016. The project duration is set to three years. The SMILE project aims at a combined research approach into a new innovative European launcher for an emerging market of small satellites up to 50 kg using a multidisciplinary design and optimisation approach strengthened by the demonstration of critical technologies for cost-effective solutions and complemented with the design of a European-based launch capability from Andøya (Norway).

Aiming for commercial launch prices of less than 50,000 €/kg up 50 kg payload capacity, the total maximum cost for a launch shall be well below 2.5 M€. This target cost drives the design, construction, and operation of the launcher. After 2020, it is anticipated that the market for launching small satellites is in the order of several hundred per year and growing. A total capacity of up to 50 launches per year is foreseen. Using a flexible configuration of the launcher-payload interface structure, several combinations of small satellites up to 50 kg can be served.

The launcher will use advances in technology to achieve cost reduction, including design for series production, reusability, and the use of COTS components. Critical technologies enabling affordable and independent access to space will be developed in this project. To be able to meet the target price, the design will be based on existing advanced technologies as a starting point, and drive the development of required new technologies forward as part of the program. The overall objectives of the SMILE project therefore are:

- To design a concept for an innovative, cost-effective European launcher for small satellites
- To design a Europe-based launch capability for small launchers, based on the evolution of the existent sounding rocket launch site at Andøya Space Center
- To increase the Technology Readiness Level (TRL) of critical technologies for low-cost European launchers
- To develop prototypes of components, demonstrating this critical technology
- To create a roadmap defining the development plan for the small satellites launcher system from a technical, operational and economical perspective

Figure 2 shows a high level system view approach for three parallel paths throughout the project. The path towards a conceptual design of the launcher is split into an architectural design phase and a detailed design phase. Likewise, the critical technology path is split into two phases: a preparation phase and a demonstration phase for developing prototypes. The ground segment depends on the launcher design, but will also supply requirements to the launcher, and its phases follow the launcher development.

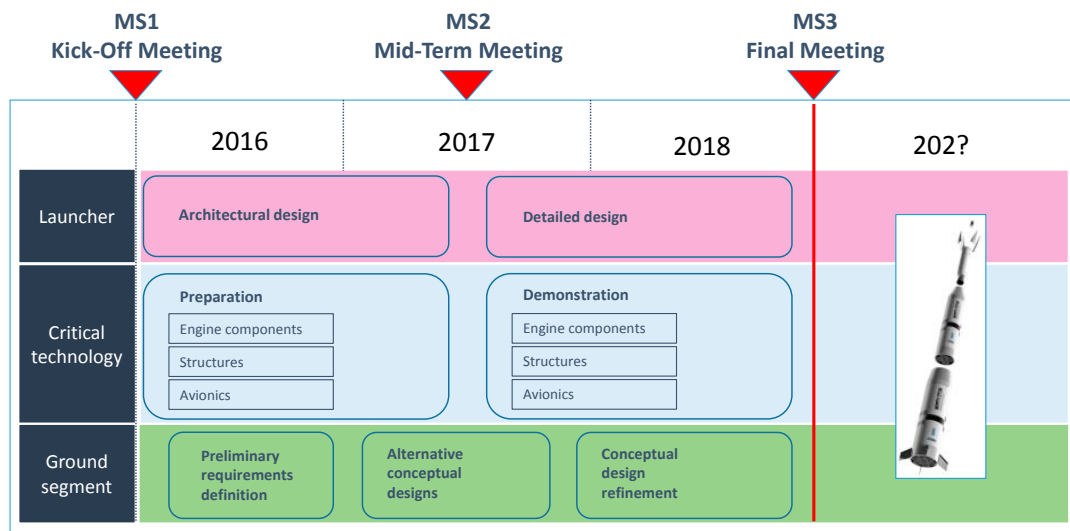


Figure 2 High level system view (source: Andøya Space Center)

In order to fulfil the project's objectives the consortium has identified a number of technologies that are capable of upgrading the actual state-of-the-art of this type of vehicles. These include:

- Hybrid engine technology
- Liquid engine technology with transpiration cooling
- Advanced low-mass and low-cost materials
- Series production of low-cost composite structures
- Printing technology for low-cost metal components
- Advanced, reliable COTS technology for miniaturised, low-power avionics
- European-based launch facility

At the end of the project the target Technology Readiness Levels (TRL) for the critical technologies shall be according to Table 1.

Table 1 Target Technology Readiness Levels (TRL) for the critical technologies

Item	TRL
Launcher concept	2
Hybrid rocket engine	7
Liquid rocket engine	5/6
Advanced materials	3
Automated manufacturing of composites	5
Printing technology	8
Advanced avionics	4
European launch facility concept	2

In order to enhance the continuity of the project's objectives, a roadmap will be set-up by assessing scenarios and critical future steps at technical, financial, and organisational levels. A business development shall include a technology roadmap towards a TRL 9 launcher. Furthermore, it presents a strategy to achieve commercially feasible launch services, including cost – benefit analysis.

Although critical technologies in several areas are encompassed by the SMILE project, the focus in this paper is on novel hybrid and liquid rocket engine technologies by Nammo Raufoss AS and the German Aerospace Center DLR respectively. Especially, the paper addresses the needs and impacts of these technologies on a small launcher development as well as the foreseen necessary costs reduction. In SMILE the following objectives are foreseen for critical engine technology development:

- To perform a trade-off between two propulsion technologies in order to obtain the configuration answering the best to the constraints of the project
- To design the architecture of the launcher's propulsion modules based on the requirements
- To generate the detailed design of the propulsion modules
- To select technology for low-cost advanced engine parts
- To produce prototypes of the selected engine parts
- To conduct firing tests of the liquid engine

3 Hybrid Rocket Engine Technology

3.1 Current State of the Technology

Up to now, only two kinds of engines have been used for operational launchers: liquid engines (such as the European Vulcain II, the Russian RD-180 or the American Merlin 1A) and solid engines. The latter are mainly used as boosters for the big launchers (Ariane 5's SRBs) or for the first stages of medium launcher (Vega's P80, Pegasus system) or sounding rockets.

Liquid engines offer high versatility, through thrust regulation and restart capabilities, and high performance (high specific impulse), but are somewhat limited in thrust and their high complexity (with a turbo-pump feeding the combustion chamber with propellants) makes them quite costly, both in terms of mass budget and development cost. On the other side, while solid engines offer simplicity and high performances in terms of thrust, they have the drawbacks of being inherently hazardous (the oxidizer and fuel are intimately mixed in the grain), uncontrollable (impossible to stop once ignited), and tailored to one specific task.

Hybrid propulsion development started at the same time as for the other two. The goal was to combine the advantages of both types of engine (inherent safety, versatility, being able to throttle, and simplicity) at low cost. Unfortunately, knowledge at that time didn't allow hybrid engines to compete in terms of performance, especially because of a low regression rate of the fuel (leading to only small thrust capabilities, or complex fuel grain geometry).

In the last decade, however, hybrid propulsion has matured, mainly through research and technology programs. Full scale flight weight rocket motors are now totally conceivable at low price, and with capabilities and performance allowing a competition with liquid or solid engine.

Nammo Raufoss AS (Nammo), a Norwegian based defence company, has since 2003 invested in the hybrid rocket propulsion technology. Based on hydrogen peroxide (H_2O_2), a completely green oxidizer, and HTPB fuel, Nammo has moved the technology forward through the following projects:

- The upscaling of the hybrid technology to a 30kN-class engine under the ESA funded Future Launcher Preparatory Program (FLPP)
- The establishment of a new 500kN Green Propulsion Test Stand
- The development of a throttleable hybrid engine for a Lunar Lander under the European Community funded 7th Framework Program, SPARTAN
- The development of a so-called "Hot Gas Reaction System" (HGRS), a new (mono-propellant) Reaction Control System for Ariane 5ME, Ariane 6 and Vega to replace the hydrazine alternative

The combination H_2O_2 /HTPB offer the advantage of being already available in industrial quantities, while being completely green (only CO_2 and H_2O produced), safe to handle (nontoxic products) and safe to operate (two propellant completely segregated). Those characteristics, coupled with a simple fluid system, will substantially reduce hybrid propulsion life cycle cost compared to other propulsion systems. Moreover, with the use of a catalyst bed to decompose the H_2O_2 , the engine can be stopped

and restarted at will, without the need of an external igniter (which is the case with liquid engine). This could prove crucial for small launchers that want to launch multiple payloads on different orbits.

With Nammo's hybrid architecture, it is possible to develop an engine with performances high enough to suit the needs of small satellites launchers, at a much lower price tag.

3.2 The Unitary Motor

The current state of the hybrid technology at Nammo is represented by the Unitary Motor (UM), a novel concept of hybrid rocket engine developed by Nammo under an ESA-FLPP contract. It uses high concentration hydrogen peroxide (87.5% H_2O_2) as oxidizer and hydroxyl-terminated polybutadiene (HTPB) rubber as fuel. Its working principle is shown in Figure 3. The incoming liquid oxidizer, with a mass flow of about 11 kg/s, is first decomposed over a catalyst into hot steam and gaseous oxygen to a temperature of 670°C . It then goes through the injector and enters the combustion chamber in hot gaseous form, where ignition of the hybrid combustion occurs without any dedicated ignition device due to the high oxidizer heat flux, sufficient to vaporize the solid fuel. The vortex flow-field in the chamber generated by the injector helps in maintaining a high heat flux to the fuel surface and in achieving appropriate mixing of the reactants for a high combustion efficiency. The hot product gases are then expelled through a nozzle, generating close to 30 kN of thrust.

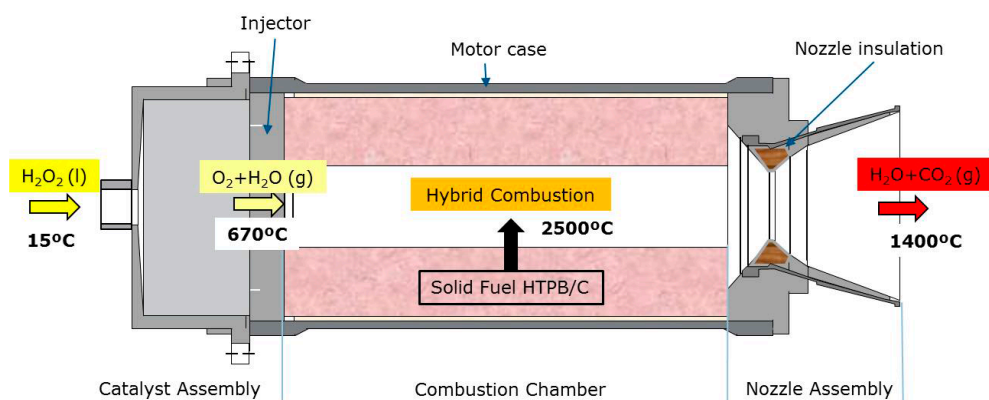


Figure 3 Working principle of the Unitary Motor

Compared with solid rocket motors, the Unitary Motor designed by Nammo has a rich set of attractive features, even when compared with other versions of hybrid rocket engines with which it shares the inherent properties of hybrid propulsion. These features are:

- Self-ignition increasing engine start reliability and enabling an unlimited restart capability
- Wide range throttling with limited performance losses
- Green life cycle and exhaust properties
- Solid inert fuel and high-density green storable oxidizer
- High engine combustion efficiency, performance and stability
- Simplicity of a single circular port and single feedline configuration
- Low development and operational costs

Some of these features are common with liquid rocket engines, but compared with liquid rocket engines, the architecture of the UM is much simpler and the same features are obtained for a fraction of the cost.

The design of the UM has been split in two phases. First, a Heavy-Wall configuration (HWUM) has been designed, manufactured and tested in the fall of 2014. The goal was to assess the up-scaling of the hybrid technology (i.e. inner ballistic, regression rate of the fuel) without the constraint of a flight-weight engine. The HWUM demonstrated great behaviour in terms of both performance and stability from the first test firing (see Figure 4 and Figure 5), and continued to do so throughout the rest of the campaign. This allowed Nammo to complete the HWUM development test campaign in only 6 hybrid firing tests and one iteration on the motor configuration. The HWUM ground tests were concluded with the delivery of a very satisfactory motor design yielding the performance desired (see Table 2) for the next stage in the program.

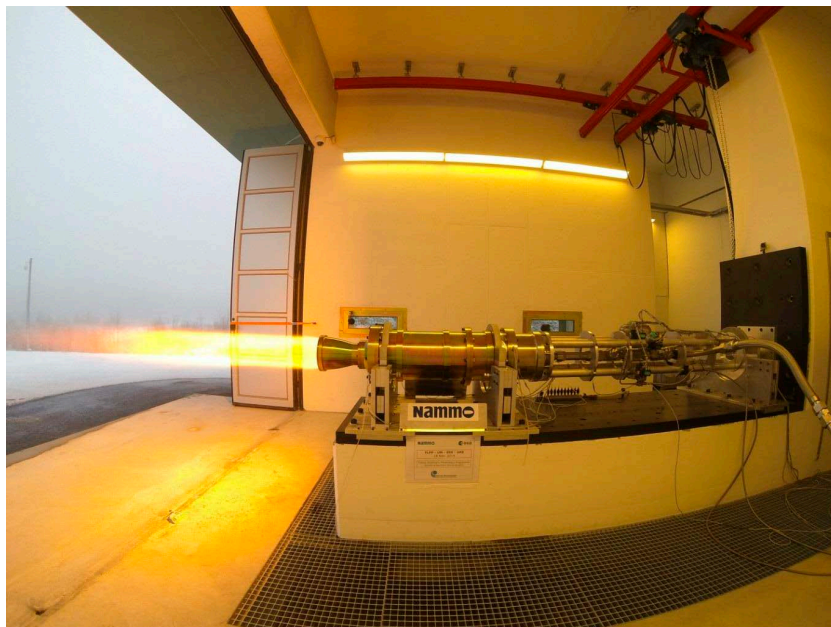


Figure 4 HWUM during 3rd firing on November 18th, 2014

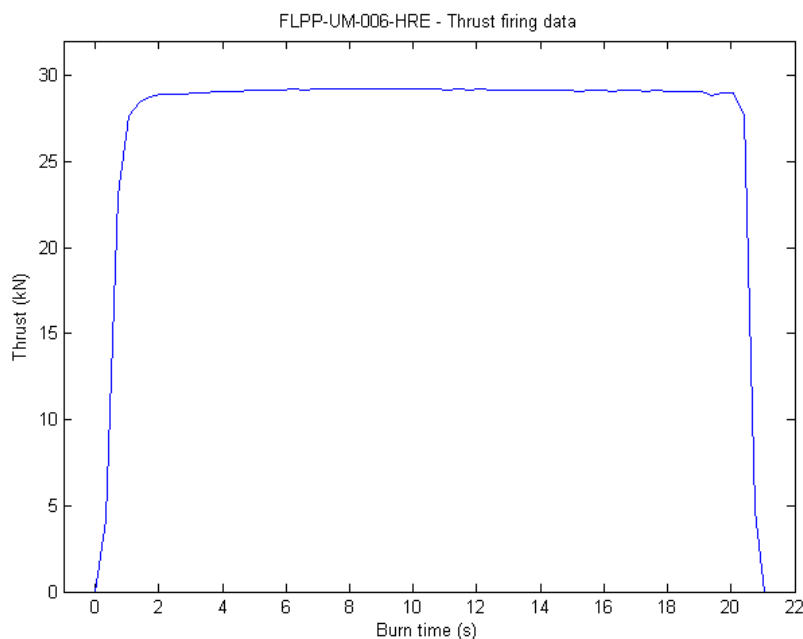


Figure 5 Thrust measured during the 4th HWUM firing, on November 28th, 2014. The measured data has been filtered down to 3 Hz sampling for visualization in this paper

Table 2 Comparison of the 5th HWUM test experimental results (December 09th, 2014) with the motor design target. In this table all mean values are averaged over the entire motor burn duration and all values have been rounded independently

Firing	FLPP-UM-007-HRE	Design model target
Burn Duration	25 s	25 s
Mean oxidizer mass flow	10.8 kg/s	10.8 kg/s
Mean fuel mass flow	1.9 kg/s	1.6 kg/s
Mean oxidizer to fuel ratio	5.75	6.75
Mean chamber pressure	36 bar	35 bar
Mean specific impulse (ground level)	234 s	230 s
Mean engine efficiency	95 %	94 %
Total impulse (ground level)	750 kNs	700 kNs

Based on the results from the Heavy Wall Unitary Motor firings, a Flight Weight Unitary Motor (FWUM) has been designed. This design is currently being manufactured and the test campaign should start in November 2015. The design of the FWUM mainly replaces over-dimensioned parts with optimized parts, but it will also increase the capabilities of the Unitary Motor. Based on discussions with the user community, the capabilities of the UM are adjusted to a larger total impulse capability of 1000 kNs approximately. Based on the demonstrated performance of the HWUM, this can be achieved within an outer diameter of 14 inches, which is the standard sounding rocket payload diameter in use at Andøya Space Center and Europe in general. The updated design data is given in Table 3. Although also feasible, no attempt has been made to achieve a higher thrust level for the FWUM, but rather a longer burn time. It is increased with 10 sec. from 25 sec. to 35 sec.

Table 3 Main differences between the HWUM and the FWUM

Property	HWUM	FWUM
Total impulse	750 kNs	980 kNs
Outer diameter	305 mm (12 in.)	356 mm (14 in.)
Burn duration	25 s	35 s
Dry mass (without consumed fuel)	>280 kg	<100 kg
Consumed fuel mass	< 50 kg	> 60 kg
Consumed oxidizer mass	~270 kg	~380 kg

A demonstration launch of the FWUM is planned for the fall 2016 on board a prototype Nucleus sounding rocket (based on a single UM) from Andøya Space Center in Northern Norway. The goal of the launch is to reach the space frontier at 100 km altitude.

3.3 Hybrid Rocket Stage for a Micro-Launcher

In order to keep the price of the propulsion system as low as possible, reusability of components is a key feature leading to cost reductions through volume production and increased reliability through automated production. In that sense, the Unitary Motor is thought of as a building block that can be clustered to deliver the required thrust for a micro-launcher. The North Star rocket family, a Norwegian initiative of sounding rockets and micro-launchers, is based on that principle, with the utilization of two high thrust motors, the UM and its future upgrade the UM2, for the first stages and a third high performance engine with a more moderate thrust requirement and longer burn-time needed to obtain orbit insertion on the upper stage. Figure 6 presents the concepts of the different rockets of the North Star Family and Figure 7 the preliminary design performance of the different propulsion stages.

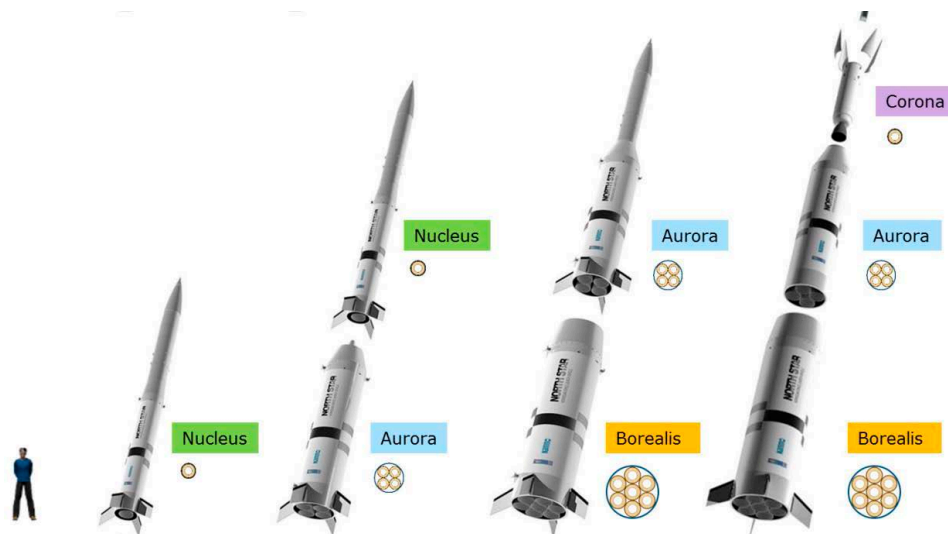






Figure 6 The North Star Rocket Family (source: Andøya Space Center)

Rocket stage	Motorization		Indicative impulse
Nucleus	1x Unitary Motor 1		Thrust: 28 kN Burn time: 35 s Total impulse: 1 MNs
Aurora	4x Unitary Motor 1		Thrust: 114 kN Burn time: 35 s Total impulse: 4 MNs
Borealis*	7x Unitary Motor 2		Thrust: 450 kN Burn time: 64 s Total impulse: 30 MNs
Corona*	1x High Performance Hybrid Motor		Thrust: 5 kN Burn time: 70 s Total impulse: 0.35 MNs

*The characteristics of the stages Borealis and Corona will depend on the flight performance of the Nucleus and later on, the Aurora stages.

Figure 7 The North Star Rocket Family (source: Andøya Space Center)

In SMILE, the same principle will be used with the added value of combining hybrid stages of clustered Unitary Motors with liquid stages. Based on the results and performances obtained during the FWUM test campaign and the demonstration launch, the sizing of the different propulsion stages of the micro-launcher will be achieved by clustering the Unitary Motor. The fluid feeding system (bringing the liquid oxidizer to the motors) will have to be design and sized accordingly and the performances (i.e. thrust, specific impulse, weight and size envelope) will be provided to the other members of the consortium for the global design of the launcher. It is strongly believed that both the inherent lower price of the hybrid technology and the clustering of elements enabling a more cost-effective production will be a large contribution in bringing the global cost of the launcher within the required range of 50,000€/kg.

4 Liquid Rocket Engine Technology

Liquid propulsion is a well proven technology that can be operated with different types of propellants. Hereby, the choice of propellants is driven by their resulting specific impulse, thrust-levels, and tankage-to-propellant mass ratios. Hence, for lower stages high-density propellants are preferred which yields into both reduced tankage volume and geometrical expansion ratio. For this reason, LOX/kerosene is rather used for first stages than LOX/LH₂; in the latter case a combination with solid boosters (e.g. Ariane 5 and Space Shuttle) would be aimed for the launch or the propellants are preferably applied to upper stages as LOX/LH₂ offers the highest specific impulse.

In general, liquid propulsion is a reliable technology which is very promising due to its flexibility as the engines can be throttled at a wide range and easily re-ignited. For the current configuration, the combination of LOX/kerosene propellants is considered as very favourable. Kerosene can be easily stored and refuelled, is a cheap fuel, and is available worldwide.

In any case, the propulsion system is the most expensive part of the launcher. Thus, it would be beneficial to retrieve the engines back after a launched mission. Possible solutions might include guided parachutes, propulsion-assisted boosters (like SpaceX), winged fly-back engines (like Adeline from Airbus Defence & Space) or winged fly-back boosters where DLR already did some studies within the FLPP programme funded by the European Space Agency (ESA). Once the engines are retrieved, they have to be inspected in order to have them refuelled and put into operation again.

4.1 Ceramic Based Design

In contrast to solid, hybrid or classical liquid engine approaches, liquid engines based on a ceramic design are very promising candidates with respect to such reusability aspects as they offer:

- Improved lifetime
- Thermo-shock resistance
- Thermal-cycling ability
- Reliability and damage tolerance
- Reduction in structural weight
- Oxidation resistance
- High specific strength at elevated temperatures
- Low thermal expansion

Hence, this specific kind of propulsion system using ceramics is well suited and applicable as it can be thermally cycled without degradation which is not the case for metallic approaches.

At DLR, there is a long experience on liquid rocket propulsion. The Institute of Structures and Design in Stuttgart is thereby focusing on ceramic-based designs which are based on the transpiration cooling technique. This is very favourable as all ceramic materials, such as non-oxide and oxide ceramic matrix composites (CMCs), can be manufactured in-house^{4, 5, 6}. The transpiration cooling principle enables to

highly increase the chamber wall lifetime while permitting a slight decrease of specific impulse. Compared to classical metallic solutions, it is possible to substantially reduce the engine's structural weight, depending on applied ceramic materials⁷ and proposed design. In general, transpiration cooling consists of two mechanisms, as depicted in Figure 8: A small portion of the coolant is penetrating the combustor walls and thereby convectively extracting heat from the hot wall; in addition, a coolant layer forms at the inner combustor wall which protects the wall from hot combustion flow.

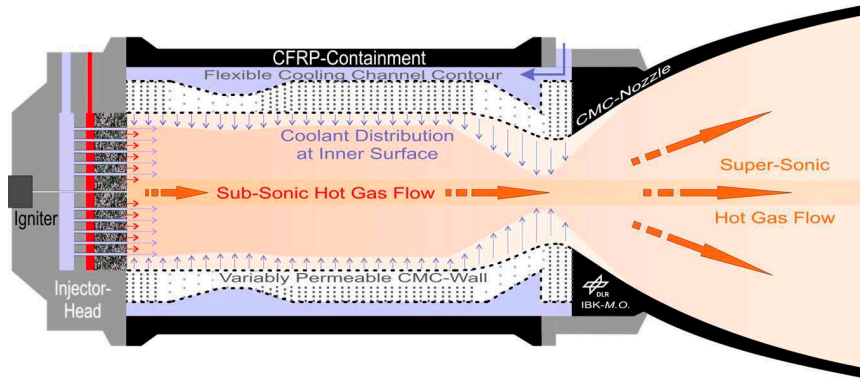


Figure 8 Schematic of a transpiration cooled ceramic thrust chamber

First initial experiments on transpiration cooled segments for liquid rocket propulsion have been performed at the end of the 1990s. All testing was performed at various high-performance rocket engine test benches of DLR Lampoldshausen, up to 90 bars combustion chamber pressure. They solely focused on hydrogen-oxygen propellants, including cryogenic conditions as well. The development resulted in sophisticated design approaches which were investigated in different projects.

Between 2008 and 2012, four separate test campaigns were performed within the DLR projects KSK (Keramische Schubkammer, ceramic thrust chamber) and KERBEROS (Keramische Bauweisen für Experimentelle Raketenantriebe von Oberstufen, Ceramic Design of Experimental Rocket Engines for Upper Stages), as given in Table 4. The different configurations included the variation of wall and nozzle materials, injectors (API: advanced porous injector from DLR Lampoldshausen; TRIK: coaxial injector by DLR Stuttgart), contraction ratio, coolant blowing ratio, characteristic chamber length, etc. Further details can be obtained from^{8, 9, 10}.

Table 4 DLR ceramic thrust chamber test campaigns 2008-2012

	KSK-KT	KSK-ST5	MT5-A	WS1
Year	2008	2010	2012	2012
Test bench	P8	P8	P6.1	P6.1
Propellant combination	LOX/LH2	LOX/LH2	LOX/GH2	LOX/GH2
Injection temperature (fuel)	≈ 55 K	≈ 55 K	≈ 135 K	≈ 150 K
Injection temperature (oxidizer)	≈ 155 K	≈ 155 K	≈ 125 K	≈ 140 K
Coolant	H ₂	H ₂	H ₂	H ₂
Wall material	C/C	Al ₂ O ₃ and C/C	Al ₂ O ₃ and C/C	Various

	KSK-KT	KSK-ST5	MT5-A	WS1
Nozzle material	Copper	C/C	C/C	C/C
Injector	API	API	TRIK	TRIK
Chamber diameter (d_c)	50 mm	50 mm	50 mm	50 mm
Throat diameter (d_t)	31.6 mm	31.6 mm	20 mm	20 mm
Characteristic chamber length (l^*)	0.86 m	0.68 m	1.75 m	1.83 m

Figure 9 shows test operation of the ceramic thrust chamber during the test campaign MT5-A. Especially in combination with the transpiration cooling technique and the use of CFRP housing structures, the engine's structural weight can be significantly reduced. On the other side, sophisticated CMC materials enable replacing ITAR-controlled metal alloys (as the current main material for combustion chambers) in the future.

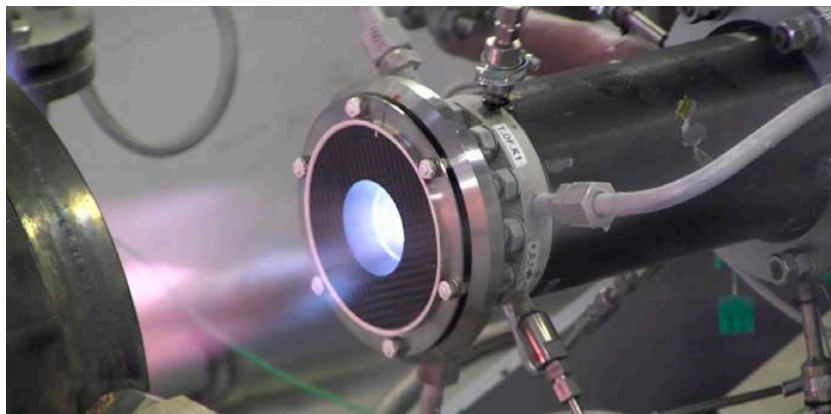


Figure 9 Liquid rocket engine test MT5-A at P6.1 test bench in Lampoldshausen (LOX/GH2)

Furthermore, the general feasibility in GOX/kerosene combustion environment was successfully demonstrated in the EC project ATLLAS (coordinated by ESA and funded within FP6, 2006-2009). All tests were performed at the high-pressure rocket combustion chamber test bench at Technische Universität München (TUM), see Figure 10. Various CMC materials were tested, whereas oxide CMCs seem to be very suited for this kind of application as the material is able to withstand hot gas oxygen attacks. Figure 11 shows two of the integrated CMC liner materials: C/C (non-oxide) and WHIPOX (oxide). With respect to cooling performance, hydrocarbon-based coolants such as Jet A-1 kerosene turned out to be very efficient.

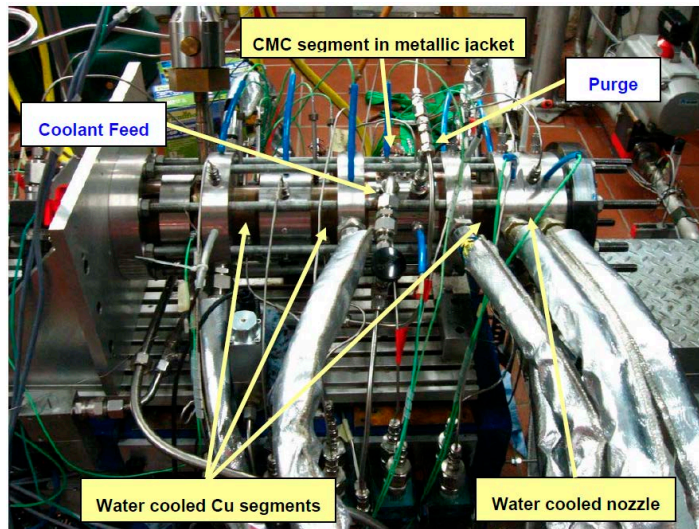


Figure 10 Liquid rocket engine test at TUM test bench (GOX/Jet A-1)



Figure 11 Ceramic inner liners for TUM test (left: C/C, right: WHIPOX)

4.2 Engine Reusability

Already in the late 1960s, Pratt & Whitney developed the transpiration cooled XLR-129 rocket engine with a chamber pressure of approximately 100 bars. The engine was extensively tested and based on the results; a transpiration cooled design was developed for the Space Shuttle Main Engine (SSME). Transpiration cooling was selected in order to fulfil the NASA criteria of 100 time engine reusability¹¹. This engine development of Pratt & Whitney is the only known experimental study dealing with transpiration cooled engine life cycle, durability and re-usability to date. Based on the published results of Pratt & Whitney and theoretical considerations, the lifetime of transpiration cooled chambers is expected to be at least 10 times higher than that of regeneratively cooled chambers.

It has to be mentioned that at this time, transpiration cooling research was mainly conducted considering metallic materials. In case of local hot spots, such metallic structures tend to melt and cause a catastrophic failure. This is in clear contrast to ceramic-based materials which do not exhibit such behaviour. Additionally, ceramic-based designs enable improved lifetimes due to their positive thermal-cycling ability and thermo-shock resistance.

4.3 Envisaged SMILE Approach

It is expected that existing design approaches could be transferred to LOX/kerosene operation. In doing so, a ceramic-based thrust chamber assembly will be designed. Whereas the injector head might be made via SLM (selective laser melting)-techniques, the combustor component will be designed of ceramic liners actively cooled by transpiration. Here, both fuel and oxidiser are considered as potential coolants. In addition, a ceramic nozzle section is foreseen.

A clustered design is considered which would result in multiple turbopump-fed sub-scaled engines, depending on the mission scenario. DLR's engine enables reliable low-cost components to fit into the envisaged target price of 50,000€ per kg of payload with a future potential of reusability.

5 Conclusions

There is a need for a dedicated and affordable small satellite launcher. A major challenge for the launcher design is to become cost efficient within all technology development areas in order to offer future customer launch prices of less than €50,000 per kg of payload. The SMILE project will take up this challenge by aiming at a combined research approach into a new innovative small launcher for an emerging market of small satellites up to 50 kg using a cost-effective design approach. Cost reduction is achieved by applying reusability of one or more stages, applying commercial industry-grade components and through volume production including cost-optimized manufacturing process. In this paper the cost effectiveness for the rocket engine development is addressed.

For the hybrid rocket engine development this is achieved by the inherent low life-cycle cost of the hybrid technology and the clustering of unitary propulsion elements, the Unitary Motor. Low life-cycle cost is achieved by a simple architecture, the non-toxicity, the inertness and the availability of the propellants and the overall low development and operational costs. The clustering of the Unitary Motor will also bring the cost down, thanks to a higher volume production for each component. This higher volume could also legitimate an automated production leading to a better reliability of the product.

For the liquid rocket engine development this is achieved by an operation of multiple LOX/kerosene sub-scaled engines based on ceramic materials and a transpiration cooling technique for improved engine lifetime and reuse. In combination with reliable low cost 3-D printed components and the potential use of CFRP (carbon-fibre reinforced plastics) housing structures, the engine's structural weight can be significantly reduced.

The combination of the two hybrid and liquid propulsion technologies will allow the use of the right technology at the right place to offer a launcher delivering the required performance at the lowest price possible. Ultimately, the choice of the propulsion system for all the stages of the rocket will be a trade-off between performance, launch objectives and cost.

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